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TECHNICAL MEMORANDUMS

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

No. 717

DEVELOPMENT OF THE RULES GOVERNING THE STRENGTH OF AIRPLANES

By H. G. Küssner and Karl Thalau

PART II

LOADING CONDITIONS IN GERMANY (CONTINUED), ENGLAND AND THE UNITED STATES

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PART II. LOADING CONDITIONS IN GERMANY (CONTINUED), ENGLAND, AND THE UNITED STATES

5. Development in Germany Since 1926

During the first few years after the war German airplane activities were practically wiped out, whereas the development in other countries progressed, particularly in the design of large aircraft engines with low specific weight. The biplane, preferred during the war, continued in favor, although improved aerodynamically, and the speeds, which toward the end of the great conflict had reached 200 km/h (125 mi./hr.) with the fastest airplanes, could now be increased considerably.

Advance in stunting had reached the stage where dives, spins, loops, barrel rolls, etc. were no longer a novelty. The stress of the airplanes had so enormously increased by the higher speed, as well as by the audacity of the flight evolutions, that the load factors customary during the war had, for pursuit airplanes, for example, been raised to more than double, in order to avoid wing failure.

When in May 1926, the restrictions were finally removed, it was necessary to completely overhaul the last official specifications, i.e., the 1918 BLV.

Even back in 1918, airplane manufacturers and others registered complaints and recommendations for modifications

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*"Die Entwicklung der Festigkeitsvorschriften für Flugzeuge von den Anfängen der Flugtechnik bis zur Gegenwart." Luftfahrtforschung, June 21, 1932, pp. 25-38. (For Part I, see N.A.C.A. Technical Memorandum No. 716.)

of the BLV, namely:

1) The specified load factors appear too high. The effect of edge and intermediate strip on the covering does not appear to exist to the extent that, for instance, in case A the 5.5 instead of the 4.5 load by minimum loading is warranted;

2) In cantilever wings the favorable influences of the plate effect are no longer great;

3) The raise in dissipating moment in case C is especially clumsy. The magnitude of the moment should likewise be graded according to stress categories;

4) The regulation governing distortion in strength tests is felt to be too severe. It is not, in this instance, a matter of simple permanent form changes, but rather of such which vitiate the aerodynamic behavior. Sand loads should only be applied so that no permanent deformations occur at one half the specified load factor. This would conform to the wishes of the manufacturers. Sand loading up to the full load factor could still be carried out, but merely for the purpose of detecting weak points in a design.

5) The choice should be restricted to statically comprehensive designs to make an exact stress analysis possible;

6) It does not seem permissible to permit the computed wing stresses to approach the breaking limit, and certainly not as far as metal design is concerned;

7) Raising the unit loading of the movable tail surfaces from 150 to 300 kg/m^2 (30.7 to 61.4 lb./sq.ft.) is incomprehensible; 200 kg/m^2 (41.0 lb./sq.ft.) would be more correct.

8) Details are awaited as to whether the energy absorption of the shock absorber may be allowed for in landing-gear strut analysis;

9) There are no data on minimum possible flight path curves for investigating asymmetrical load cases.

Items 4) and 5) were discussed at the meetings of the Standards Committee on December 12, and 19, 1918; item 6) was to be taken up later. Special regulations for commercial aircraft were to be issued.

In this connection, Rohrbach made a noteworthy proposal (reference 39). Proceeding from the assumption that an airplane is just as severely stressed in a turn as in a pull-out and that the power output is proportional to the air density, he rechecked different war airplanes and arrived at the formula

$$n_{A_{Br}} = 2.4 \frac{\rho_o}{\rho_G} \tag{20}$$

as breaking-load factor. This formula contains indirectly the wing loading, power loading, and coefficient of climb, and is therefore better suited to the particular qualities of a certain airplane type than a diagrammatical load factor. Unfortunately his premise of maximum stress in a flat turn is untenable, with the result that his formula had no practical significance.

To obtain experimental data for loading conditions, a program of acceleration measurements in flight with a number of modern airplane types was undertaken by J. v. Köppen in 1926 and 1927. The accelerograph, developed by H. Wendroth and G. Wollé in the D.V.L., recorded the accelerations on a blackened drum. The recorded load factors are reproduced in table XII (reference 40). The highest possible values of the load factor $n = v^2/v_l^2$ obtainable in a sharp pull-out were almost obtained (bracketed figures in table XII).

Table XII. Load Factors Recorded in Flight

Airplane type	Junkers G 24	Dornier Komet	Albatros L 68	Albatros L 68 a	Junkers A 20	Junkers A 35	Dietrich DP 9
Wing area (m^2)	89	62	21.8	24.4	29	29	14
Gross weight in test (kg)	(5000)	(2500)	730	797	1400	1420	(600)
Horsepower	690	360	70	100	230	300	50
Landing speed V_L (km/h)	90	100	65	75	80	80	65
Top speed (km/h)	178	180	147	137	185	206	135
Turn	1.6 to 1.8	1.8 to 2.0	2.3 to 2.7	2.7	2.8	2.5	-
Spiral	-	-	3.8	2.9	3.1	3.0	-
Spin	-	-	3.0 to 3.5	2.4 to 2.7	3.2 to 3.6	3.5 to 3.6	-
Barrel roll	-	-	4.0	2.7	4.1	3.5	3.9
Loop	-	-	2.8 to 3.8	2.6 to 3.0	3.0 to 3.4	2.9 to 3.0	3.1
Slow pull-out at speed (km/h)	1.7 1.9 140 170	2.0 2.2 185 210	2.1 to 3.1 180	2.4 3.6 200 240	3.4 to 3.9 250	3.1 250	- -
Sharp pull-out at speed (km/h)	2.2 (2.4) 140	2.6 185	4.7 (4.7) 130	3.6 4.0 200 240	- -	4.2 (5.1) 180	3.8(4) 130

(m^2 x 10.7639 = sq.ft.) (kg x 2.20462 = lb.) (km/h x .62137 = mi./hr.)
(m x 39.37 = in.) (m/s x 3.28083 = ft./sec.) (kg/m^2 x .204818 = lb./sq.ft.)
(kg/cm^2 x 14.2235 = lb./sq.in.) (t x 2204.62 = lb.)

D.V.L. Loading Conditions, 1926-1928

After various modifications from time to time between 1919 and 1926, the D.V.L. at last issued a preliminary report on load specifications, on October 15, 1926, drawn up by Hoff, Madelung, Thalau, and Uding. These loading conditions were prefaced by the following general rules and regulations:

1. Certain safe loading attitudes (for instance, flight, control, landing or transport attitudes) are set up. These safe (abbreviated for "required as safe") loads shall be service loads, that is, it is assumed that they occur in service.

2. The loadings are considered as static.

3. It must be proved that the structural safety against failure is at least 2,000 for every component part in the safe loading attitude.

4. Special safety regulations are given for the chassis. Other exceptions are specially noted.

5. No deformations must remain after a safe loading attitude.

6. The vibration strength of parts subject to vibrations, such as power plants, wing and control-surface fittings, shall not be exceeded in any safe flight attitude.

7. The strength factors which are based upon the stress analysis must be proved by tests.

8. Vital parts of an airplane, not amenable to exact stress analysis, must be strength-tested.

9. Statically of vital importance are such parts, the failure of which lowers the structural safety of the airplane at any point to or below half the stipulated figure.

The change from breaking load to "safe" load in the specifications had already been advocated by Rohrbach,

back in 1918, and was incorporated in the Holland specifications of 1924.

The 1926 D.V.L. loading conditions carried five stress categories, namely:

1. Special purpose;
2. Freight carrying;
3. Commercial;
4. School and training;
5. Acrobatic.

The four load cases of the BLV were supplemented by the E case. Position and direction of the resultant of the air loads are no longer defined, but depend upon the wing polar. The basis is the A case with extremely forward c.p. by lift coefficient c_{a_A}, which, however, need not correspond to the maximum lift. The "safe" dynamic pressure

$$q_A = \frac{n_A}{c_{a_A}} \frac{G}{F} \tag{21}$$

is derived from the "safe" load factor n_A. Originally, the corresponding safe speed with safety factor 2, according to equation (18), was $v_a \sim \sqrt{1.5/2}\, v_h = 0.87\, v_h \sim v_r$, so that in a pull-out at cruising speed v_r, there actually was a 2-time breaking safety relative to the highest possible air loads. With v_l as minimum floating speed (landing speed) the safe load factor in case A is, on the other hand

$$n_A \sim \frac{v_a^2}{v_l^2} \sim \frac{v_r^2}{v_l^2} \tag{22}$$

since in both attitudes, pull-out and landing, about the same high lift coefficient prevails. Although, according to that, the safe load factor n_A is dependent upon the speed range $v_r:v_l$, it was nevertheless decided for the above reasons of simplicity, to introduce constants for n_A which agree with the numbers of the categories (with the exception of category 1). Thus, for category 2, it is

$n_A = 2$, for category 3, $n_A = 3$, etc. This was based on the assumption that the stresses hinge substantially upon the skillful manipulation of the controls and that a well-trained pilot would not go beyond the stipulated load factors. All other loading conditions were derived from load factor n_A and the safe dynamic pressure q_A. It is in

Case B: $n_B \sim 0.67\ n_A$, $q_B = 3\ q_A$, $c_{a_B} = 0.22\ c_{a_A}$

" C: $q_C = 4\ q_A$, $c_{a_C} = 0$

" D: $n_D \sim 0.33\ n_A$, $q_D = 3\ q_A$, $c_{a_D} = -0.11\ c_{a_A}$

" E: $n_E = 0.5\ n_A$, $q_E = 1.5\ q_A$, $c_{a_E} = -0.33\ c_{a_A}$

Should the negative lift coefficient $c_{a\ min} = c_{a_E}$ be known, it is permissible to put

$$q_E = \frac{n_E}{c_{a_E}} \frac{G}{F} \text{ in case E, and}$$

$$q_D = 2\ q_E, \quad c_{a_D} = 0.33\ c_{a_E} \text{ in case D}$$

The load distribution is the same as in the 1918 BLV, but may also be assumed conformably to aerodynamic experiments, in which case the aileron effect must be taken into account. The torsion of the wings shall not exceed 3.5° at any point under the effect of the "safe" dynamic pressure in the C case.

The horizontal tail surfaces must withstand the loads produced by the moment equalization in all flight cases A to E, and besides the safe additive moment

$$M_H = 0.02\ q_B\ F^{1.5} \tag{23}$$

in case B. The vertical tail surfaces shall be designed to withstand the safe moment

$$M_s = 0.015\ q_B\ F^{1.5} \tag{24}$$

The mean safe aileron loading is computed from the pressure distribution in cases B and C. This loading is, in case B, to be combined with a safe aileron moment about the longitudinal airplane axis

$$M_Q = 0.015\ q_S\ F^{1.5} \tag{25}$$

These additive moments are assumed as originating from the formulas

$$\left.\begin{aligned} M_H &= c_{n_H}\ q_B\ F_H\ l_H \\ M_S &= c_{n_S}\ q_B\ F_S\ l_S \\ M_Q &\sim \Delta\ c_a\ q_R\ \frac{t_R}{t}\ F_Q\ l_Q \end{aligned}\right\} \tag{26}$$

The normal force coefficients were assumed at $c_{n_H} = 0.20$ and $c_{n_S} = 0.25$, and the lift increment of the wing portion ahead of the ailerons at $c_a = \pm 0.65$. Statistical data on 42 German airplanes disclose as average

$$F_H\ l_H \sim 0.10\ F^{1.5}$$

$$F_S\ l_S \sim 0.06\ F^{1.5}$$

$$\frac{t_R}{t}\ F_Q\ l_B \sim 0.023\ F^{1.5}$$

The purpose of these formulas was to oblige to a certain minimum fuselage strength and, indirectly, a minimum tail surface size, because the tails were habitually designed too small, as result of the unit loadings specified in the BLV. It was fully recognized by those who formulated this specification that, influencing the size of the tail could be much better effected by strictly aerodynamic stability regulations, but unfortunately, the necessary experimental foundations were not available at that time.

The strength of all control parts shall at least be for 40 kg (88.2 lb.) safe handling, and for 150 kg (330.7 lb.) safe foot power from all possible directions.

The load distribution on the tail surfaces in chord direction shall, lacking model tests, be assumed 1) triangular, and 2) rectangular. No control surface - by locked stick - shall show more than 7.5 percent of total displacement under safe load. The increase in control force due to friction shall not exceed 20 percent.

The required energy absorption of the landing gear shall suffice for the sinking speed

$$w = 0.141\ v_l \quad \text{in category } 2,3$$
$$0.155\ v_l \quad \text{"} \quad \text{"} \quad 4$$
$$0.127\ v_l \quad \text{"} \quad \text{"} \quad 5$$

Half of the corresponding impact energy shall be absorbed by the shock absorber, thus insuring the safe impact factor $e \geqslant 3$. The initial tension shall not exceed the airplane weight at the most. The safety factors against impact are:

1.20 for axle or equivalent parts,

1.60 for wheels and rest of landing gear,

2.00 for remainder of airplane.

The other half of the stipulated energy absorption can be covered by work of deformation of the axle or equivalent parts. The safe tail skid impact factor e_{sp} shall be analyzed from its energy absorption. The support pressure of the tail skid on the ground divided by the acceleration due to gravity is figured as effective mass. Here it would be more precise to use the reduced mass computed from the moment of inertia. The landing gear shall be analyzed for

1a. three-point landing, impact factor e;

1b. wheel landing, resultant passing through the center of gravity, impact factor e;

2. one-wheel landing, impact factor 0.33 e;

3. lateral landing, impact parallel to wheel axle, impact factor 0.10 e;

4. combination of 1 and 3, also 2 and 3, each with 75 percent of the individual loads;

5. landing impact from the front in horizontal direction, impact factor 0.67 e, only in combination with 1 and 2 each with 75 percent of the individual loads;

6a. nosing over, impact factor 2.5;

6b. dragging of tail skid, impact factor 1.5, resultant sloping 15° forward;

7. horizontal impact on tail skid from the side, impact factor 0.1 e_{sp}.

Fuselages shall be analyzed for maximum flight and landing stresses. By simultaneous application of horizontal and vertical tail-surface loads, both shall by analyzed at 75 percent. At landing of flying boats a safe load of 5.0 G is to be evenly distributed over that part of the hull bottom which first contacts with the water (zone of step). The loading conditions for float supports are as shown in figure 35.

The engine mounting must, aside from the cited fuselage stresses, be able to withstand

1. in flight, the static thrust, the maximum engine torque, and gyroscopic moment of the propeller in combination;

2. oblique landings at 10° sidewise and forward;

3. by damaged propeller, a centrifugal force of

$$Z = 0.00015\ D\ n^2\ m_s.$$

The installation of fuel tanks and equipment shall be analyzed at 25 percent greater safety than the other parts. The safe load on seats and safety belts shall be 200 kg (440.9 lb.), vertically and horizontally.

Despite the many objectionable features of this first preliminary draft of the D.V.L., which were due to some extent to the unfortunate interruption in airplane design and research, they still represent a notable contribution to the development of loading conditions and were freely drawn upon in the revision of several foreign load specifications. In the second draft, of August 25, 1927, the definitions are sharper and some factors modified. But the most important change is the grading of the load factors according to total airplane weight. The simple grading, according to purpose of use had proved insufficient for categories 1 to 3. But in order to maintain the classification according to purpose groups, the relationship to gross weight G was given in terms of empirical formulas, namely:

$$\left.\begin{array}{ll} \text{Group 1,} & n_A = 1.6 + 1{,}000:(G + 1{,}500), \\ \text{" 2,} & n_A = 1.8 + 1{,}000:(G + 1{,}500), \\ \text{" 3,} & n_A = 2.0 + 2{,}000:(G + 2{,}000). \end{array}\right\} \qquad (27)$$

The smallest obtainable radius of curvature in a turn or pull-out depends chiefly on the fuselage length, i.e., the distance of the control surfaces from the c.g., and on the elevator displacement, which frequently is synonymous with the ratio of control force of the pilot to the gross weight of the airplane. Since this interdependence is neither theoretically nor experimentally explained, the load factors were approximately assumed contingent upon the gross weight, on the premises of stated flying speeds.

Motivated by the 1918 BLV, practically all subsequent loading conditions in Germany, as well as in all other countries, reveal this crude dependence of the load factor on the gross weight of the airplane in one form or another.

On the motion of airplane manufacturers, the load factor of groups 2 and 3 (commercial airplanes) was not inconsiderably lowered, a step which later was bound to prove of questionable merit in the face of the consistently greater speed range, as it left the original assumption $v_a \sim v_r$ more and more behind.

The manufacturers also caused the modifications of the specifications for vertical tail surfaces and ailerons to be made as follows:

The vertical tail surfaces and the ailerons shall be strong enough to sustain the 'safe' moment $M_s = 0.012\ q_B\ F^{1.5}$, thus reducing the corresponding normal force and lift coefficients to $c_{ns} = 0.20$ and $\Delta c_a = 0.52$.

Unless servo-motors are used, the strength of the rudder bars, transmission cables, control lever with support shall be analyzed as follows:

a) elevator controls, 50 kg "safe" manual force by symmetrically applied force;

b) aileron control, 25 kg "safe" manual force by stick control, and a "safe" turning moment of 40 d by wheel control;

c) rudder controls, 50 kg, one-sided acting "safe" foot power, 100 kg on either side in rough landing.

Cases a) and b) shall be in combination with three fourths of the individual loads. This demand was, however, voided in 1928. By dual control the assumption of simultaneous operation of both controls with three fourths of the individual loads is specified.

The manual forces in airplane controls formed the subject of an elaborate report by H. Hertel (reference 41).

The sinking speed for analyzing the shock absorption of the landing gear was lowered to

$$w = 0.110\ v_l, \quad \text{for group 1,}$$

$$w = 0.127\ v_l, \quad \text{" groups 2 and 3,}$$

$$w = 0.141\ v_l, \quad \text{" group 4,}$$

$$w = 0.110\ v_l, \quad \text{" " 5.}$$

The safety factors for the wheel axle were raised to 1.6, and for the wheels and the rest of the landing gear, to 1.8. (See fig. 36.)

The loading conditions for float-support systems were recently revised by H. Ebner, while Lewe (reference 42) and Bottomley (reference 43) published an account of measurements and observations made on seaplanes during the war.

On the basis of these data, two formulas were developed for the impact factor, i.e., the ratio of impact to dead weight, and specifically for

$$\text{Seaway} \leqslant 3 : e \doteq 0.256\ \frac{v_l^{1.5}}{G^{0.25}}$$

$$\text{"} \quad > 3 : e = 0.350\ \frac{v_l^{1.5}}{G^{0.25}}$$

These formulas represent a compromise insofar as the landing speed for flat hull bottoms is theoretically linearly, and for V-shaped bottoms quadratically expressed in the formulas (reference 43), while the exponent of G is still

dependent upon the type of enlargement.

The impact is classified according to the following load cases (see fig. 37):

1a. Nose landing, impact at 30 to 45° to horizontal passing through the c.g., impact factor e;

1b. Step landing, impact normally through the c.g., impact factor e;

1c. Stern and two-wave landing, vertical impact, impact factor e;

2. One-sided landing on one float or one half of hull, impact factor 0.5 e;

3. Side landing, impact horizontal at lower hull edge, applied front or rear, impact factor 0.33 e.

And in the following combinations: (1) + (3) with 75 percent each; (2) + (3) with 100 percent each.

The initial discrepancies in the load cases for float seaplanes and flying boats were later removed. These strength specifications made no claim to anything complete or definite, and were from time to time amended and revised with the cooperation of the airplane manufacturers.

In the third draft of the loading conditions of February 27, 1928, issued in connection with new Design Specifications of the D.V.L. for Airplanes, the factor of safety was reduced to 1.8. But a safety factor of 1.4 against exceeding the 0.2 limit (yield limit) was set up in place of it. In addition, in load testing brand new airplanes with 1.26 times safe load no form changes exceeding 5 percent of the total deformation shall remain.

The purpose of these two rules was to avoid permanent deformations under "safe" load by unsuitable structural material. Since the yield limit for most airplane metals lies at about two thirds of the breaking stress, the parts designed for stress failure have as a rule at least twice the breaking strength. The parts designed for stability failure are more favorably situated in this respect inasmuch as they remain intact by stresses up to near ultimate load and are more resistant to dynamic stresses and local

damages; for which reason the safety factor for these parts could be lowered to 1.8.

In prolonged vibration attitudes combined with static stresses, a 1.25 times safety against fatigue failure is demanded. But this specification has not been applicable heretofore, owing to the difficulty of defining the initial stress and the vibration amplitudes in service. Besides, it was intended more to focus the attention of the designers on the actual service conditions, with a view to impressing it on their minds in future designs.

For groups 4 and 5, the safe load factor in case A was raised from 4 to 4.5 and from 5 to 6, so as to insure at least the old ultimate load factor with the new 1.8 factor of safety. Ostensibly, stresses higher than the safe loads were held more probable in the training and acrobatic airplanes of groups 4 and 5 than in the others. (See the Holland Specifications, Part III of this report.) The lift coefficient c_{a_E} for the E case shall always be taken from the wing polar.

For wings which in the "safe" C-case moment reveal more than 3.5° distortion, mathematical proof must be adduced to show that no wing flutter can occur.

The sinking speed which decides the energy absorption of the landing gear is

$w = 0.077\ v_l$, for group 1,

$w = 0.095\ v_l$, " groups 2 and 5,

$w = 0.105\ v_l$, " group 3,

$w = 0.118\ v_l$, " " 4,

and the factors of safety are:

$S_a = 1.45$ to 1.55, for axle and shock-absorbing parts;

$S_b = 1.55$ " 1.65, for wheels, chassis and skid;

$S_c = 1.8$, for remainder of airplane;

$S_d = 2.3$, stroke stop.

When S_a, S_b are raised, then S_c, S_d must be raised in the same proportion. The load cases for the landing gear are illustrated in figure 38.

The safe loading for fuselages in nosing over shall be analyzed at 1.5 G, the breaking strength of the forward parts up to the passenger cabins shall not be more than 1.55.

The regulations for float supports are extended conformably to figure 39. The "safe" impact factor is

$$e = C_0 \; C_1 \; \frac{1 + a}{1 + a + a^2} \; v_l^{1.5} \qquad (28)$$

with $a = 0.178 \; G^{0.25}$ and $C_0 = 0.055$ and 0.072 in seaways 3 to 5, respectively.

The effect of the Vee is allowed for by factor $C_1 = 1 - 0.7 \cos \beta/2$ in the individual load cases. The maximum pressure on the bottom is obtained when the step impact loads raised by 50 percent are distributed over 20 percent of the float area. The effect of special bottom design on the impact factor may be allowed for in each case. Some pertinent investigations on stresses in flying boats in landing have been published by Pabst (D.V.L.) (reference 45), H. Wagner (reference 46), and Taub (reference 47).

The Work of the German Aircraft Committee (DLA) During 1930 and 1931

This third draft of loading conditions was, aside from minor changes, in force until the summer of 1930. In the spring of the same year the DLA took charge of all further development work on airplane loading conditions. The section "Loading Conditions," of which the writers are respectively, secretary and collaborator, has now begun a gradual revision of the third draft, which is issued in loose-leaf form.

The most urgent problem was the reinsertion of the flight speed into the loading conditions, because the hypotheses on the speed range forming the basis of the preceding drafts, had not held step with the advance made in

aviation. In order to avoid such a recurrence in the future, it was decided to use simple formulas, say, on the order of equations (6) and (7), which are amenable to mechanically similar interpretation rather than constant load factors.

The most frequent flight attitude, when estimating the stresses of other than acrobatic airplanes, is that of cruising flight. The cruising speed is never exactly known, although it may be assumed at around 85 percent of the maximum level-flight speed v_h by 15 percent specified engine throttle, which can be computed and measured reasonably accurately. Since occasionally full throttle is also used in cruising, the dynamic pressure q_h for maximum unaccelerated horizontal flight was provisionally chosen as criterion of the stress.

Leaflet No. 1, released by the DLA in the fall of 1930, incorporated the following amendments:

> The safe dynamic pressure in case C (dive) shall, at the highest, be equal (always equal for group 5) to the obtainable final dynamic pressure, but may, for groups 1 to 4, become
>
> $$q_c \geqq q_h + \Delta \quad \text{or} \quad \geqq 2.25\ q_h$$
>
> Groups 1 and 2, $\Delta = 200\ \text{kg/m}^2$,
>
> Group 3, $= 250$ "
>
> " 4, $= 400$ "

Hereby it is assumed that, intentionally or unintentionally, a steep glide with loss of height of $\Delta h > \Delta : \gamma$ from level flight can be executed.

It must be proved that wing flutter cannot occur below the speed

$$v_k = 1.3 \sqrt{\frac{2\ q_c}{\rho}} \tag{29}$$

(reference 48).

The air density is

$\rho = 0.12,$ for groups 1 to 3,

$\rho = 1.2\ \rho_G \leqq 0.12,$ for groups 4 and 5.

This rule was laid down as substitute for the previous restriction of wing torsion angle in case G and is a criterion for the necessary torsional stiffness of the wings.

The dynamic pressures in load cases B and D are:

$$q_B = q_D = 0.8\ q_C.$$

The loading conditions of the third draft of the DVL were supplemented by

1. Load case G (gust stresses). The safe load factor is:

$$n_G = 1 \pm \frac{1}{16}\ v_h\ w\ \frac{F}{G}\ c_a{}'\ U_o \tag{30}$$

$w = 10$ m/s vertical velocity of air current;

$U_o \sim 2/3 \leq 1$ coefficient of gust stress (reference 49)

$c_a{}'$ change of lift coefficient of wing with angle of attack.

2. Load case H (zooming over obstacles). The safe load factor is:

$$n_H = 1 + \frac{v_h^2}{gr} \leq \frac{F\ q_h\ c_{a\ max}}{G} \tag{31}$$

The radius of curvature of the flight path is

$r = 300 + 0.01\ G$ for group 2,

$r = 250 + 0.01\ G$ " " 3.

3. Allowance for asymmetrical loads.

One half of the wing is assumed to be loaded with 1.0, the other half with 0.7 of the safe load factor (case C excepted). The load distribution of the wing halves shall correspond to that of the load cases. The moment of the air load resulting from the vertical tail-surface loading about the normal axis shall, with a

view to the mass forces of the fuselage and of the wings, be in combination with load cases B and D.

The peculiar feature of load case G is that the pilot has little or no influence on the height of the gust stresses insofar as his flight schedule and cruising speed are concerned; for which reason this loading condition is so eminently fitted as minimum requirement for commercial aircraft.

The provisional establishment of a gust rise of 10 m/s is not free of arbitrariness. Owing to insufficient knowledge of individual factors of product $C = \frac{\rho}{2} w c_a' U_0$, together with the fact that the constructive data relative to the meteorological data, have less effect on the stress, the recheck of product C from the breaking strength of proved airplanes is preferable for a start.

Next to load case G, a combination of cases G and B by greater flight speed must eventually be investigated also, because several wing failures within the last few years, which proved fatal to experienced pilots, are presumably due to a moderate initial acceleration in gliding superposed by an undecided violent gust.

Our lack of knowledge on the expectancy of gust intensities of definite magnitude, as well as of the structure of gusts, offers therefore a profitable field of exploration for the immediate future.

The load factor of case H corresponds to Reissner's formula of 1912. The pull-out radius r, depends, as previously stated, primarily on fuselage length, airplane gross weight and control force, then on the psychological effects on the pilot, as, for instance, effect of fright when suddenly face to face with the ground, or an obstacle. The figures for r, while admittedly plenty high, are intended as guide until more accurate information is available.

Load cases G and H are to take the place of the old load cases A and D later on. It probably will then also be possible to set up a simple, continuous design specification for the whole range of lift coefficients from 0 to $c_{a\,max}$, from which the particular loading conditions

are to be selected for a given wing cellule. (See fig. 42, Part III.)

Assuming the wings to be self-lifting in flight, the wing stress is approximately dependent upon the mass force $K = n\ G_R$ exerted upon the fuselage. With the load factor n = constant, as in accepted practice hitherto, the stress is apparently proportional to the body weight G_R. This holds good only for the left side of equation (31) with the newly laid-down cases G and H, whereas the right side of equation (31), which after all corresponds to equation (22), as well as equation (30), manifests wing stresses which are no longer proportional to the body weight G_R. When the strength of an airplane does not comply with the conditions (30) and (31), a reduction in fuselage weight by restriction of useful load, as customary up to now in such a case, has relatively little effect on the wing stress. More effective although not always feasible, is the removal of the useful load from the fuselage and into the wing. Usually the conditions (30) and (31) are difficult to comply with when the airplane is designed for large power excess, which, on one hand, is a very desirable feature from the safety point of view, on the other hand, can be used occasionally to obtain high speed by opening to full throttle. When the unnecessary use of high speed is prohibited as is customary in many other modes of transportation, the requirements for wing strength can be moderated without thereby impairing the usefulness of the airplane.

The pilots must be made to understand that high speed constitutes a source of danger as in all other modern methods of transportation, a fact which incidental to the rules of constant load factors and the confidence placed in them, is occasionally forgotten.

Leaflet No. 2, released in February 1931, embodied the following amendments:

The stability limits by static load (column effect, buckling) shall not be exceeded at 1.8 times the safe load.

The breaking stress by static load (tension, compression, flexion) shall not be exceeded at 2.0 times the safe load.

The 0.2 limit shall not be exceeded at 1.35 times the safe load when metal is used as structural material.

For welded parts the strength characteristics in the annealed zone shall be used, unless the structural part is subsequently age-hardened.

When structural parts are statically load-tested the breaking load is that load which the part can support for at least one minute before failing.

In statically vital connections additional stresses which may occur in service must be provided for by appropriate over dimensioning (friction, clearance, etc.).

Vitally important parts of the airplane, whose strength can not be accurately analyzed mathematically, shall be load-tested. A part is vitally important when its failure under safe loading attitudes might lead to other failures.

The fatigue strength shall not be exceeded by 1.35 times the amount of frequent stress reversals.

Whereas the stability failure of a wing cellule is comparatively reliably calculable or experimentally reproducible and little affected by local material defects, the tension failure is largely dependent upon such defects which may equally be produced in service. It would therefore be risky to make the failure of the whole wing cellule contingent upon such an unreliable quality of the structural material as the tearing strength is. Inasmuch as in light constructions the parts designed for stability failure or those with a view to the manufacture are by far in the majority, the 11 percent strengthening of the members designed for failure in tension specified above results in slightly greater weight but it makes in return the prediction of the test engineer on the ultimate strength so much more independent of the very unreliable tearing strength.

Occasioned by an accident due to failure of the vertical tail surfaces, leaflet No. 3 was released in August 1931. It contains the following amendment:

The safe mean unit loading of the vertical tail surfaces for all airplanes shall be

$$p = \frac{\rho_o v_h w}{2} \frac{dc_n}{d\alpha} \tag{32}$$

The gust velocity is, as before, $w = 10$ m/s.

The effect factor is put at $U_0 = 1$, because owing to the high inertia moment of the airplane about its normal axis, the vertical tail surfaces yield only gradually to the gust and may also be impressed by an additional stress due to elevator displacement.

The close of 1931 found a number of other recommendations submitted by specialists which were to clear away various subject and editorial matter in the released loading conditions and which are briefly summarized hereafter.

To facilitate the issuance of strength specifications, it was found necessary to set up transitory rules. Airplanes now in service or under construction shall be permitted to fall short 15 percent of the provisionsal load quota, provided proof is given of 5,000 flying hours of service without damage of the particular airplane type up to 25 percent. Interpolation by Gauss' error curve is permitted. These deficiencies, however, are not permissible for continuous indubitably known stresses.

The modifications incorporated in leaflet No. 2 shall be explained by footnotes and in their ambit of use be restricted in conformity with the state of the technique. The gust stress of wings shall be investigated not only for full useful load but likewise for flight without useful load. Load case H in leaflet No. 1 shall be superseded by the specification of a minimum dynamic pressure in case A. It shall be

$$q_{A\ failure} > q_h$$

The determination of the pressure distribution in span and wing-chord direction shall be left to the designer more than heretofore, since he has ample theoretical and experimental data to guide him (reference 80, Part III, N.A.C.A. Technical Memorandum No. 718). The horizontal tail surfaces also shall be analyzed for gust stress, but with the effect factor which corresponds to the airplane mass reduced to the c.p. of the tail surfaces and the downwash factors 0.8 and 0.9 for biplanes and monoplanes, respectively.

Furthermore, horizontal and vertical tail surfaces shall be so designed as to be able to support a safe stress by elevator manipulation with the normal force coefficient $c_n = 0.5$ and the dynamic pressure q_h, airplanes of groups 4 and 5 up to a dynamic pressure of $6\ q_{min}$. These stresses

shall be combined with those effected by the static moment balance. In addition to that, one half the stress of the vertical tail surfaces by gusts or elevator displacement shall be combined with half the stress produced when a wing engine fails. The mean safe aileron loading is found from a flight attitude with dynamic pressure q_h, together with an aileron deflection of about 15° and such a uniform rotation of the airplane about its fore and aft axis that the total lift of one wing half does not change relative to undisturbed flight. In order to guard against wing flutter the mean torsional stiffness of the overhang of the wings and of the tail shall not be less than

$$D_o = \frac{\rho_o}{\rho} q_c F_o^2 \qquad (33)$$

with $\rho = 0.12$ for airplanes of group 1 to 3, and air density ρ for groups 4 and 5, at which the dynamic pressure q_c is quickest obtained after a dive from the service ceiling level. To support the pilot in a rough landing, a safe foot power of 150 kg each on both sides of the foot pedals shall be assumed.

The landing gear shall be designed to withstand a safe impact of 0.7 e G which so passes behind the c.g. as to form an angle of 20° with the line connecting wheel center and center of gravity. By application of the brake on the ground the friction coefficient 0.5 between the ground and the undeflected tire shall be assumed.

A braked three-point landing with a normal force of 0.8 e G acting on the landing gear and the friction coefficient 0.3 shall be analyzed. By braked turning on the ground the sum of the safe lateral mutually parallel wheel and tail-skid pressures shall be 0.5 G.

The regulations for landing gears with skids are the same as for those fitted with wheel brakes. In the event that the skids are exclusively used for snow or ice landing, the frictional forces may be ignored.

The unit loading of snow skis on the ground shall not exceed 1,000 kg/m².

The nose of the fuselage shall be designed to withstand, when both wheels touch the ground simultaneously at 60° to the horizontal, a rearward directed safe impact $K = b\ m_R$, whereby the deceleration is $b = 15\ m/s^2$.

Minimum requirements shall be prescribed for the torsional stiffness of the tail-carrying fuselage end, so as to avoid the buffeting produced by eddies from the wings and fuselage.

In addition, it was planned to apply the laws of eccentric impact to all those load cases of the landing gears and float supports in which the stress is substantially dependent upon the amount of inertia moment.

Loading Conditions for Gliders

In view of the almost 1,500 gliders in Germany as compared to 900 airplanes, it was deemed fitting to include the specifications for gliders and sailplanes as formulated by the Research Institute of the Rhön-Rossitten Society (reference 50).

Table XIII. Load Factors for Gliders, 1927

	Flight Stresses							Landing stress Wings, fuselage with fittings	
	Wings				Tail surfaces breaking load	Control			
	Normal flight		Dive			Highest load on control stick			
	n	S	n	S	kg/m²	kg	S	n	S
Hanging glider	3	1	1	0.5	150	--	-	-	-
Seat glider	3	2	1	1	150	50	3	4	2
Training glider	3	2	1	1.5	300	50	3	4	2
Performance glider	3	2	1	1	150	50	2	3	2
Experimental glider	3	2	1	2	300	50	2	4	2

In this table (13) n represents the required factors which practically correspond to the "safe" load factor in the 1926 DVL loading conditions.

The stress analysis shall be so drawn up that in every case the maximum material stresses are determined, so that

the safety factors are

$$S = \frac{\sigma_{\text{permissible}}}{\sigma_{\text{calculated}}}$$

permissible must be proved by data on material.

The load distribution over wings and control surfaces shall be effected conformably to their ground plan form.

In multiplanes a higher loading of the upper wing must be allowed for.

The normal flight position is the flight with minimum sinking speed.

The turning moment of the wing in diving with terminal velocity is:

$$M = G_R \; t \; \frac{c_{m_0}}{c_{w_0}} \tag{34}$$

Diving with terminal velocity being impossible with hanging gliders, the safety factor S, more properly load factor n, was lowered to 0.5.

Double safety shall be proved against pure frontal pressure loading in diving flight.

With a view to the danger of wing flutter the natural frequency of oscillations for cantilever wings shall be no less than 100 to 120/min.

In 1930, the Technical Committee of the Rhön-Glider contests in collaboration with W. Coulmann, K. Haarmann, A. Holtmann, A. Lippisch, E. Pfister and M. v. Pilgrim issued the "directions for the design of gliders and sailplanes" which contained the following loading conditions:

Wing Cellule

Case 1.- C.G. in extreme forward position. The application of the load is given by the c.p. position. The ultimate load factor for all groups is $n = 6$.

Case 2.- Diving by maximum torsional stress. The maximum turning moment to be used as basis of the analysis occurs when

$$\frac{c_m}{c_r} - \frac{z}{t}\frac{c_n}{c_r} \tag{35}$$

attains its maximum. The corresponding dynamic pressure is

$$q = \frac{G}{c_r F} \tag{36}$$

Case 3.- Landing stress. For the analysis the wing weight, assumedly evenly distributed over the wing, constitutes the load. The ultimate load factor is $n = 8$. For performance gliders the load factor $n = 6$ may be used provided ample skid elasticity is available.

Control Surfaces

Horizontal and vertical tail surfaces shall be analyzed with a mean loading of $p = 150$ kg/m². The total load is $Q = p\ F_H$. The load distribution over the elevator chord is rectangular over the stabilizer when the elevators are damped, triangular over the adjoining elevator, and triangular with the maximum value over the leading edge for pendulum elevators.

The ailerons shall be analyzed with a 75 kg/m² load by triangular lift distribution.

Fuselage

The fuselage shall be analyzed for bending by horizontal tail surface load and for bending and torsion by vertical tail surface load. The load is applied in the c.g. of the control surface. The neck of the fuselage is stressed by a separate load $P = 50$ kg in the plane of the wing, applied at the wing tip and facing to the rear. The take-off (or release) hook shall be calculated for a 1,000 kg pull.

The rate of vibration of cantilever and semi-cantilever wings shall be at least 120/min.

The limitation to two principal loading conditions is representative and appears applicable to airplanes also somewhat according to the French Bureau Veritas specifications (see Part III, fig. 42). At least it would be possible to deduce for monoplanes all systematic loads of the wing cellule from two loading conditions, for example, of pure loading in bending and pure torsion and compute there-

from the maximum forces decisive for the dimensioning.

Since the wing loading of gliders ranges between 8 and 15 kg/m^2 only and the minimum gliding angle must be small in order to make gliding possible, a comparatively large increase in flight speed is attainable even by a moderate push on the stick. Allowance for possible high stresses due to gusts or starting maneuvers is necessary, particularly for high-performance gliders, because the heights reached now become greater and greater and even clouds are occasionally penetrated.

6. English Loading Conditions

On January 6, 1920, the load factors subcommittee of the British Advisory Committee for Aeronautics prepared the following schedule (reference 51). This schedule was not intended as something definitely established but rather which would be revised from time to time in order that it would remain in accordance with the demands arising from improvements in the constructional methods and design.

The airplanes are divided into two classes only, the general group, which should be sufficiently strong to allow of stunt flying of all descriptions, and the commercial group, in which stunting and diving is prohibited.

The factor of safety is the quotient "breaking strength of a structural component divided by the highest possible loading in any attitude of flight," the breaking load factor is the quotient "breaking strength of a structural component divided by the loading in steady horizontal flight."

The breaking load factor n_a, in table XIV, applies in the case where the center of pressure of the wing is in the extreme forward position.

The breaking load factor n_b applies to the center of pressure in the position corresponding to maximum horizontal speed at ground level.

The breaking strength of the vertical tail surfaces shall correspond to a loading with the dynamic pressure for maximum horizontal speed v_h at ground level and the lift coefficient c_{as} of the vertical tail surfaces. (See table XIV). For the prediction of these speeds the following

dimensionless formulas are used:

For single engine aircraft,

$$\left(v_h \sqrt{\frac{\rho F}{G}} - 1.79\right)^2 = 3.16 \left(\frac{N}{G} \sqrt{\frac{\rho F}{G}} - 0.316\right) \qquad (37)$$

For multi-engine aircraft,

$$\left(v_h \sqrt{\frac{\rho F}{G}} - 1.64\right)^2 = 2.99 \left(\frac{N}{G} \sqrt{\frac{\rho F}{G}} - 0.31\right) \qquad (38)$$

For flying boats,

$$\left(v_h \sqrt{\frac{\rho F}{G}} - 1.425\right)^2 = 2.74 \left(\frac{N}{G} \sqrt{\frac{\rho F}{G}} - 0.304\right) \qquad (39)$$

The safety factor S_c is valid for the stress in diving with terminal velocity. The braking effect of the propeller may be allowed for when calculating the terminal dynamic pressure. The wing moment to be absorbed by the horizontal tail surface loading shall be, for wing section R.A.F. 15 and R.A.F. 6 C,

0.6 Gt and 0.9 Gt in single engine airplanes,
0.55 Gt and 0.7 Gt in multi-engine airplanes,
0.5 Gt and 0.5 Gt in flying boats.

The impact factor e in table XIV is decisive for the breaking strength of the landing gear, and applies to the cases of wheel landing with the chord of the wings horizontal and to three-point landing.

The impact factor is 0.5 e for oblique landing so that one wing tip touches the ground while the chord of the wings is horizontal and when the tail skid and landing gear touch the ground simultaneously. The shock absorption of the landing gear shall in all cases correspond to the sinking speed 3.05 m/s.

Table XIV. English Load Factors, 1920

Weight (t)	General class		Commercial		
	≤ 1.36	≥ 4.54	≤ 2.27	4.54	≥ 13.6
n_a	8	6	6 (5.5)	5 (4.5)	4
n_b	6	4.5	4.5 (4)	3.75(3.25)	3
c_{a_S}	1.2	1.2	1.0	1.0	1.0
S_c	1.75	1.75	1.75(1.25)	1.75(1.25)	1.75(1.25)
e	8	6	6	5	4

Note: The bracketed figures apply to airplanes which are longitudinally stable within the whole angle of attack range of normal flight. Linear interpolation is necessary between all given figures.

The ommission of wing stress from above (D-case) is noteworthy and is contrary to the experience in the other countries. Such stresses occur not only in the admittedly rare inverted flight but also in normal flight attitude by control maneuvers and gusts.

Subsequent accidents in England however caused the inclusion of this load case first for acrobatic, then for commercial airplanes also.

The English Commercial Air Law C.A. Form 17, of 1922, contained strength specifications which did not differ very much from the preceding ones (reference 52).

The schedule of load cases is as follows (see table XV):

Case a: c.p. in most forward position of horizontal flight, propeller drag included:

1. With engine off,
2. With twice the normal propeller thrust and torque.

Case b: c.p. in position corresponding to maximum horizontal speed at ground level.

Case c: Vertical dive at terminal velocity. The propeller drag may be allowed for.

Case d: The specific breaking load of the vertical tail surfaces and fuselage shall be $p = 3.92\ q_{min}$.

Case e: In landing with horizontal wing chord and in three-point landing the energy absorption of the landing gear shall correspond to a sinking speed of

$$w = 0.92 + 0.1\ v_l \tag{40}$$

Hereby the safety factor is 1.15 for the undercarriage and 1.25 for the other members of the structure.

Subsequently the division of the stress groups was amended conformably to the C.I.N.A. (International Commission for Air Navigation) recommendations to read:

1. Normal group (formerly commercial),
2. Special group,
3. Acrobatic group (formerly general group).

Table XV. English Load Factors, 1922

Weight (t)	General group			Commercial group			
	$\leqslant$ 1.13	2.27	$\geqslant$ 4.54	$\leqslant$ 1.13	2.27	4.54	$\geqslant$ 13.6
n_a	7.5	7.	6	5.5	5	4	4
n_b	5.5	5	4.5	4	4	3.25	3
S_c	1.5	1.5	1.5	1.25	1.25	1.25	1.25

The special group was intended for racing or record airplanes and had in the cases a, b and c the breaking load factors $n_a = 4$, $n_b = 3$ and the factor of safety $S_c = 1$. The fins and rudders were to be designed for a breaking load of 146 kg/m².

These specifications were superseded in 1929 by the "Air Publication 1208" for civil aircraft (reference 53), which contained the following modified breaking load factors. (See table XVI).

The strength specifications are governed by the intended purpose to which the aircraft is put. There are no definite rules for the special class (S). In case of failure of one bracing member half the load factors of cases a and b shall be sustained. The case d (inverted flight) for category A is newly introduced.

Table XVI. English Load Factors, 1929-1931

Weight (t)	Group N, commercial				Group A, acrobatic		
	$\leqslant 1.13$	2.27	4.54	$\geqslant 13.6$	$\leqslant 1.13$	2.27	$\geqslant 4.54$
n_a	5.5	5	4	4	7.5	7	6
n_b	4	4	3.25	3	5.5	5	4.5
S_c	1.25	1.25	1.2 (1.0)	1.0	1.5	1.5	1.5
S_c'	2.5	2	1.5	1.5	-	-	-
n_d	-	-	-	-	5	4.67	4

The load factor for stalled flight attitude (pancaking with slotted wings) is $n_a' = 2/3\ u_a$.

The fuselage shall sustain the following loads:

1. Pull-out at full throttle: n_a times gravity force in combinations with twice the propeller thrust, torque and gyroscopic moment by angular velocity

$$w = \frac{g\ n_{as}}{v_{as}} \quad \text{of the airplane} \qquad (41)$$

   For the engine mount $n_a \geqslant 6$ is required. The propeller thrust shall be analyzed for the "safe" case a - dynamic pressure q_{as}, i.e. at half the case a - dynamic pressure at failure.

2. Normal flight with engine throttled: n_a times gravity force, engine mount: $n_a \geqslant 6$,

3. Static propeller thrust and torque by deceleration on the ground with two times breaking load factor.

4. Inverted flight for category a.

5. Up-load on tail skid in landing with consideration to relieving mass effects. Load factor 4.5.

6. Down-load on the tail in a limiting nose dive with the safety factor of case c.

7. Side-load on fin and rudder (fuselage distortion) under breaking load $P = 3.92\ F_s\ q_a$.

In addition, the following cases shall be considered for the tailplane and elevators:

8. Up-load in normal flight, c.p. forward, load factor n_a.

9. Down-load with $P = 3.14\ F_H\ q_a$ breaking load.

The landing gear shall be capable of withstanding a vertical velocity $w = 0.92 + 0.1\ v_l$ by 1.15 breaking load factor. But the impact factor shall not be less than 4 by simultaneous touching of the wheels; the load factor on the remainder of the structure to be 1.25.

Cases to be considered are: wheel landing with horizontal lateral and longitudinal axis, three-point landing, side load with breaking load factor 0.7, and lastly, the case 3 enumerated above.

For landing wheels the normal load on a wheel base shall be $G_n \leq 9{,}100$ Db, the breaking load $P \geq 5\ G_n$. The wheel shall be capable of withstanding a load inclined at 15° and 30° to the ground with 0.52 and 0.48 P, without permanent distortion. The energy absorption by 4.2 at tire pressure shall be $1.27\ P^2 : \sqrt{Db}$ without undue permanent distortion.

The strength of all control operating systems shall be as follows (less than 400 kg weight empty in brackets): push or pull on the top of the control column 85 (42.5) kg, tangential force on hand wheel 42.5 (22.5) kg, side load on the top of the control column 42.5 kg, a push of 170 (85) kg on one side of the rudder bar, a simultaneous push of 90 kg on each side of the rudder bar. For the purpose of analysis, the efficiency of a welded joint taking tensile loads shall be taken at 0.66 and for compressive loads, at 0.75 to 1.0 depending on the circumstances in which the joint is used. The stress in tension members under theoretical breaking load shall not exceed the yield limit.

Apart from the specifications for landing wheels, the English loading conditions fail to reveal anything radically new. They approach in many ways the American specifications of 1928. It should be noted that the same term: load factor is used for load factor and breaking safety. The cases, 1, 3, 7 and 9, contain safety factor 2 in the same sense as the German loading conditions.

In May and July 1931 these specifications were amended to read as follows:

Instead of diving with terminal dynamic pressure, a glide with dynamic pressure,

$$q_c' \geqslant 2.25\ q_h \geqslant 9\ q_a \tag{42}$$

may be investigated. In which case the safety factors S_c' given in table XVI shall be used. In addition, the airplane shall be capable of withstanding a downward diverted vertical gust of 6.1 m/s by a flight speed

$$v \geqslant v_h \geqslant 2\ v_l$$

that is, with full useful load. Obviously it was taken for granted that the wing strength always suffices for sustaining an upward gust. Tailplane and elevators shall in such a glide be in equilibrium with a wing moment 0.2 Gt with the safety factors S_c'. The downward breaking load on tailplane and elevators shall be $P = 3.92\ F_H\ q_a$ conformably to that for fins and rudders.

Under these conditions, the shock absorption of the airplane wheels shall be

$$E = 2.55\ P^2 : P_R \sqrt{Db}$$

7. American Loading Conditions

For the period of from 1920 to 1922 the airplanes of the U. S. Air Corps were governed by the breaking load factors appended in table 18 (reference 54). Load cases a to c are identical with the British. In case d (inverted flight) the c.p. lies at 1/4 of the wing chord; n_R denotes the breaking load factor of the fuselage for stress analysis.

For case c, a load factor of 3 is specified for monoplanes and multiplanes without incidence bracing.

These load factors are based upon acceleration measurements in flight carried out by F. H. Norton and E. T. Allen at the Langley Field Station of the National Advisory Committee for Aeronautics (reference 55). The maximum load factor in flight recorded on the rapidly obsolescent types of airplanes of that date, was 4.2. Accordingly, the requirements for the most highly stressed type, the pursuit plane, were $2 \times 4.2 \sim 8.5$ as breaking load factor for pull-out under the assumption of 2 times breaking safety.

Toward the end of 1922, a number of wing failures, unquestionably caused by insufficient strength in the wing design, brought about an increase in maximum breaking load factor from 8.5 to 12, which it was hoped to be able to modify again later. But J. H. Doolittle's tests, continued from the fall of 1923 to March 1924 (reference 56) confirmed the justification of the severe requirements, as a glance at table XVII reveals.

It is noteworthy that the factors in pull-out measured on the Fokker PW 7 (D XII) amounted to

$$n \sim 0.95 \frac{v^2}{v_l^2}$$

and in view of the always inexact determination of the minimum floating speed v_l practically corresponded with the highest possible stress (attainment of maximum lift coefficient). The Fokker D XII is very maneuverable in combat; the elevators are well balanced. Its theoretical breaking load factor is 8.5, although it actually may be 10. Hence wing failure is feasible in pull-out at 283 km/h speed. The U. S. Air Corps also specifies that a modern pursuit airplane should be capable of withstanding quick recovery from long dives.

In the Handbook of Instructions for Airplane Designers, 1923 edition, the higher load factors, tabulated in table XVIII, are specified. The stress categories are simply divided according to purpose of use.

The wing loading p is assumed as constant in span direction, but tapering linearly from the outboard strut or a wing chord from the wing tip to 0.6 p for analyzing the inside bays and to 0.8 p for calculating the overhang.

The c.g. of the airplane shall lie between 1/4 and 1/3 of the wing chord.

The strength of the fuselage shall correspond to the breaking load of the horizontal tail surfaces. The top of the fuselage shall be designed for a compressive load of no less than 75 percent of the tension load.

The load distribution over the individual wings for biplane and triplane combination is treated in detail.

In load case 1 (a) the center of pressure on seven airfoils is at from 27 to 32 percent of the wing chord; in load case 2 (b) for thin wings at around 50 and, and for thick wings at about 60 percent of the wing chord.

The angle between wing chord and ground, with airplane resting on the ground, shall be 12 to 17°, the angle between engine axis and line connecting c.g., wheel axis, 73 to 78°. The tread of the landing gear must be no less than 0.93 of the height of the c.g. above the ground.

Fittings shall be designed for a strength 15 percent in excess of the nominal strength of the parts which they connect.

The maximum stick force is assumed as 90 kg. The stick shall be able to withstand 135 kg longitudinally and 68 kg transversely. A load of 114 kg shall be applied at the wheel rim.

Rudder bars and pedals shall be designed for 135 kg. load and for a free movement of no less than 20° in any direction, and of 3° for the stabilizer. Control friction shall be kept at a minimum.

Overhanging aileron balances are not permitted at speeds above 177 km/h.

Table XVII. Load Factors Recorded in Flight

	Single-seat pursuit			Training	2-seat observation		Bomber
Airplane type	Fokker PW 7	Curtiss PW 8	Boeing PW 9	Vought VE 9	Curtiss X 01	Douglas X 02	Curtiss NBS 4
Engine power (hp.)	420	420	420	-	400	400	800
Gross weight in test (kg)	1450	-	-	-	-	-	-
Maximum level speed V_h (km/h)	266	-	-	-	-	-	-
Landing speed V_L (km/h)	92	-	-	-	-	-	-
Wide loop	2.7	3.2	3.0	3.8	3.3.	2.6	-
Tight loop	6.1	4.3	6.2	4.4	-	2.7	-
Tail spin, power off	2.6	3.6	3.7	2.5	-	3.3	-
Tail spin, power on	2.3	4.0	3.0	2.6	3.3	-	-
Sharp vertical bank, 180°	5.7	4.0	5.9	3.2	3.5	3.3	2.0
Side slip	7.2	6.0	6.7	-	4.5	3.5	-
Sharp pull-out 95-130 km/h	1.6	2.0	-	2.6	-	-	2.1
" " " 130-160 "	2.7	3.7	3.2	4.0	-	-	2.4
" " " 160-190 "	3.9	5.0	3.9	5.0	-	-	2.8
" " " 190-225 "	5.3	5.9	5.0	-	-	-	-
" " " 225-260 "	6.4	7.0	6.4	-	4.1	-	-
" " " 260-290 "	7.8	-	-	-	5.4	-	-
Gradual " " 190-225 "	-	4.4	4.6	3.4	-	-	-
" " " 225-260 "	-	4.8	5.5	-	-	-	-
" " " 260-290 "	-	5.6	5.8	-	-	-	-
" " " 290-320 "	-	6.3	7.3	-	-	-	-
Immelmann turn	4.4	-	-	-	-	-	-
Spiral	5.5	-	-	1.4	-	-	-
Inverted flight	1.3	-	-	-	-	-	-
Sham battle	-	-	5.0	-	-	-	-
Landing and taking off	-	4.6	2.5	2.3	1.6	2.0	-

Table XVIII. Load Factors of U. S. Army Airplanes

	1920-1922							March 1923						
	n_a	n_b	S_c	n_d	p_H kg/m²	p_S kg/m²	n_R	n_a	n_b	S_c	n_d	p_H kg/m²	p_S kg/m²	n_R
1. Single-seat pursuit for day work	8.5	5.5	1.75	3.5	170	146	7	12	6.5	1.75	4.0	170	150	7
2. Single-seat pursuit for night work	7.5	5.0	1.75	3.5	146	122	6	12	6.5	1.75	4.0	170	150	7
3. Single-seat pursuit (armored)	7.3	4.5	1.75	3.0	146	122	6	10	5.3	1.75	4.0	150	120	6
4. Two-seat pursuit	7.5	5.0	1.75	3.5	146	122	6	11	6.0	1.75	4.0	170	150	7
5. Three-seat (armored)	7.3	4.5	-	3.0	122	98	5	7	4.5	-	3.0	120	100	5
6. Two-seat infantry liaison	6.3	4.0	-	3.0	122	98	5	6	4.0	-	3.0	120	100	5
7. Two-seat night observation	6.5	4.5	-	3.0	122	98	5	6.5	4.5	-	3.0	120	100	5
8. Artillery observation	6.5	4.5	-	3.0	122	98	5	6.5	4.5	-	3.0	120	100	5
9. Two-seat corps observation	6.5	4.5	-	3.0	122	98	5	8.5	5.5	1.75	3.5	150	120	6
10. Day bomber	5.5	3.5	-	2.5	122	98	5	5.5	3.5	-	2.5	120	100	5
11. Night bomber (short distance)	4.5	3.0	-	2.5	98	73	4	4.5	3.5	-	2.5	100	75	4
12. Night bomber (long distance)	4.0	2.5	-	2.0	73	49	3	4.0	2.5	-	2.0	75	50	3
13. Training	8.0	5.5	1.75	3.5	170	146	7	8.0	5.5	1.75	3.5	170	150	7

Table XIX. Impact Factors of Landing Gear

Group	Struts	Axle	Windings	Side Load
13	7.5	7	6.5	1.5
1, 3	6	5.5	5.0	1.5
2	7	6.5	6.0	1.5
4, 5, 6, 9	5	4.5	4.5	1.2
7	6	5.5	5.0	1.2
8, 10, 11	4.5	4.25	4.0	0.8
12	4	3.75	3.5	0.8

Table XX. Strength of Airplane Wheels

Tires	26x3	28x4	30x5	32x6	36x8	44x10	54x12
Axle diameter (m)	.66	.71	.76	.81	.91	1.12	1.37
Width (m)	.076	.102	.127	.153	.203	.254	.306
Working load (kg)	362	724	908	1130	1900	2940	4540
Breaking load (kg)	2950	3400	4540	6120	8160	15000	17200

The specified breaking load and operating load of the wheels at 4.6 inflation pressure is included in table XX.

These specifications were revised in the Aeronautic Safety Code of the Bureau of Standards, New York, October 1925.

The schedule sets up, as practiced in England, two categories: general and commercial, but only for airplanes up to 3.4 tons (7,500 pounds) weight. The latter includes all airplanes used for transport purposes, which are not to be maneuvered violently. Airplanes weighing up to 1.59 tons (3,500 pounds) may be admitted to the commercial class when

$$\frac{G}{N}\sqrt{\frac{G}{F}} > 39.6 \ (\mathrm{kg}\ \tfrac{1}{2}\ \mathrm{S/m^2})$$

The other class embraces all other types, including military airplanes. The required load factors are shown in table XXI.

The breaking-load factors applied to the ribs shall be at least 10 per cent greater than those specified in the table. The distribution of load between the wings, along the span and along the chord shall be based on reliable wind-tunnel tests.

The stress analysis of the wing-truss structure shall be made for three conditions with allowance for column effect, redundancies, fabric and veneer, fixity, torsion of wing, wings with more than two spars and rib stress analysis.

The load on stabilizers and elevators is p_H and $\frac{pn_a}{3}$ respectively, the balancing reactions shall be applied at the wing spars. The load on fins and rudders is $0.75\ p_H$.

For fuselage stress analysis a 5.5 kg/hp. propeller thrust and to twice the full-load engine torque shall be assumed; the factor of safety is $S = 1 + \frac{n_a}{4}$.

In crashed landings the dynamic load factor parallel to the longitudinal axis of the fuselage or at any angle up to 30° to it, shall be $e = 10$; this applies only to single-engine airplanes having a total weight of less than 2.72 tons (6,000 pounds). A dynamic load factor of $e = 15$ shall be allowed for any reduction in section modulus by the cutting-out of cockpits or by any other change in form or extent of the skin. For airplanes having a total weight of over 2.72 tons (6,000 pounds) this shall apply only to cockpits and cabins not reserved for the use of the pilot and the crew.

Several methods for fuselage analysis are given which refer to trussed fuselage, monocoque fuselage, application of column formulas, engine support and torsion members.

The ailerons shall be designed for $\frac{pn_a}{3}$ unit surface loading. On fixed surfaces, (stabilizers and fins), the load shall be taken as uniformly distributed. On movable surfaces the intensity of loading shall be taken as varying uniformly from a maximum at the leading edge to one third

The shock-absorber stops must not be reached prior to failure of shock-absorber cords, axle, or wheels.

The landing-gear struts shall be stronger than the axle.

The structure supporting the floats in seaplanes shall withstand the following loads:

1. Step load perpendicular to the thrust line, $e_1 = 8$;

2. Bow load, 4 : 1 slope toward the thrust line, $e_2 = 8.25$;

3. Side load, $e_3 = 1.8$, combined with case 1, and specifically $0.9\ e_1 = 7.2$.

The U. S. specifications conform in general to the B.L.V. regulations, but reveal much higher breaking load factors in specific cases, a direct result, as already pointed out, of the higher speed of the more modern airplanes.

In 1928 the Department of Commerce promulgated the "Requirements for Approved Type Certificates" (Civil)(reference 57) which, in particular, modified the 1925 specifications for commercial aircraft.

Figure 40 shows the load factors for the load case of high angle of attack (A case) plotted against the corresponding weight and performance loading. The curves are to be used direct for land and float seaplanes. For flying boats and amphibians having a gross weight of 2,265 kg (5,000 lb.) or less the 2,265 kg gross weight factors are to be used.

For multi-engine airplanes the horsepower to be used for computing the power loading shall be that necessary to maintain level flight with full load at sea level.

The load factors for airplanes with gross weights between 1,132 and 5,662 kg (2,500 and 12,500 lb.) shall be determined by interpolation between the appropriate curves. (See table XXI).

of that amount at the trailing edge. All members directly affected by redundant members shall be designed for a load at least 35 percent higher than that specified for the control surface as a whole.

The control-operating system shall be designed to withstand twice the tail surface load. Additional loads are:

Load on stick, longitudinally,	of	± 113 kg	(250 lb.),
" " " , transversely,	"	± 91 kg	(200 lb.),
" " wheel, longitudinally,	"	± 136 kg	(300 lb.),
" applied at rim of wheel,	"	± 127 kg	(280 lb.),
" on a rudder bar,	"	± 136 kg	(300 lb.).

The mechanism for adjusting a stabilizer or other similar slow-motion irreversible control mechanisms shall be designed to carry a load corresponding directly to that which the surface itself is required to carry. Wherever there are redundant members in the control system, the total load must be 30 percent higher.

Analysis of the landing gear includes the following combinations of loads:

1. A vertical drop with its thrust line horizontal and with velocity at impact w given in table XXI. Static proof test of every new design should be made by a drop from 75 percent of the height specified in the table;

2. A vertical drop, but only with 0.9 velocity at impact and 8° inclination, three fourths of the energy at striking being absorbed in one wheel if there is no equalization mechanism available. If the necessary data for this calculation are not available, the transverse component of the load may be taken as one fourth of the vertical;

3. A vertical drop, but with 0.97 w velocity at impact, the thrust line sloping 12° forward;

4. Three-point landing with 0.87 w velocity at impact.

Table XXI. U. S. Load Factors, 1925

Class	Commercial			General				
Weight (t)	< 1.36	1.36-2.04	2.04-3.4	< 1.36	1.36-2.04	2.04-3.4	3.4-5.9	> 5.9
Case a	6.5	5.8	5.2	10.	8.5	6.5	4.8	4
Case b	4.5	4.0	3.6	6	5	4.2	3.3	2.8
Case c	1.5	-	-	2	1.5	1.5	-	-
Case d	2.5	2.2	2.0	4	3.3	2.7	1.8	1.6
$\frac{p_H}{v_h}$	2.79	2.45	3.12	3.57	3.01	1.9	1.34	1.11
$p_{H\ min}$	97.6	78.1	68.3	122.1	102.5	85	58.6	48.8
h (m)	0.60	0.76	0.91	0.43	0.55	0.67	0.91	0.91
w (m/s)	3.36	3.83	4.24	2.89	3.29	3.63	4.24	4.24

In the low angle of attack condition (B case) the load factor shall be 0.65 of the high angle of attack factor, but in no case less than 3. For the inverted flight and nose-dive conditions, the load factor shall be 0.40 of the high angle of attack factor, but in no case less than 2.

In a nose-dive condition the beam loads on the front spars are the same as the design breaking loads in inverted flight. By taking moments about the rear spars, it is possible to determine the load required on the tail surfaces, to maintain equilibrium. Since the sum of the forces parallel to the beam direction must be zero, if the airplane is to be in equilibrium, the total upward load on the rear spars will equal the sum of the downward loads on the front spars and tail surfaces. These loads are distributed in the usual manner. The chord components of the air loads on the wings shall be proportional to the chord loads in the low angle of attack condition, but their total magnitude shall equal the gross weight of the airplane.

In addition, there are numerous specifications for load distribution over the wings, control surfaces, and spars.

At the wing tip a linear drop in unit loading p from 0.5 to 0.8 p is assumed, while at the same time the bending moments of the overhang of braced wings shall be increased 30 percent and the loads in the center struts and brace wires by 15 percent.

Table XXII. Landing Stress and Control Surface Loading

Class gross weight (t)	Dynamic load factor e landing	Height of drop for shock-absorber design (m)	Breaking load	
			aileron tailplane elevators (kg/m²)	fin and rudder (kg/m²)
< 1.13	6.5	0.61	147	110
2.27	5.5	0.56	122	92
5.63	5.0	0.51	98	74
> 5.63	4.5	0.46	98	74

The landing gear shall be analyzed for level landing condition. The resultant passes vertically through the wheel axles and the center of gravity; its vertical com- is e $(G - G_L)$.

This loading condition shall be combined with a side load of one fourth the vertical component on both wheels. The required impact factor is $e/2$ and the course of the reactions in fuselage and wings should be examined. The three-point landing condition is also included.

For airplanes equipped with brakes the coefficient of friction is 0.55, and the stress in braked three-point landing attitude with impact factor $e/2$ should be investigated. The landing gear shock absorber shall be designed to absorb the energy corresponding to the free drop listed in table XXII without, however, subjecting the gear to forces greater than those corresponding to the load factors in landing. The tail-skid shock absorber shall be designed to resist a free drop in the three-point landing attitude equal to that listed in table XXII, without imposing loads on the skid or fuselage greater than those corresponding to the load factors for landing. If a shock absorber designed to dissipate the energy of the free drop by the flow of a liquid through an orifice (the oleo or oleo-pneumatic type of absorber) is used, the load factors required in the design of the landing gear may be reduced not to exceed 25 percent. When this type of shock absorber is used, suitable provision must be made to carry the shocks due to taxying after the shock absorber has been forced to the full extent of its stroke. No reduction may be made on the required height of free drop for the shock absorber.

In seaplane landing with horizontal propeller axis the resultant is, first, inclined so that its horizontal component is equal to one quarter of its vertical component; second, inclined so that its lateral component is equal to one quarter; third, perpendicular through the center of gravity. The impact factor of the vertical component (gross weight of airplane less weight of floats and float bracing) is 8.

Flying boat and seaplane float hulls shall be designed to maintain without permanent set a load of 0.56 kg/cm^2 (8 lb./sq.in.) over that portion of the hull lying between the first step and a section 25 percent of the distance between the step and the bow; a load of 0.28 kg/cm^2 (4 lb./sq.in.) from that section to a section at 75 percent of the distance between the step and the bow; 0.28 kg/cm^2 (4 lb./sq.in.) for the section between the first and second step, or between the step and a section halfway between the hull aft of the step.

For the remaining parts of seaplanes, with the exception of floats and fuselage attachments, the load factors are the same as for landplanes. (See table XXII.) Wing fittings shall be designed for a load 20 percent higher than the design load of the parts to which they are attached.

Streamline wires shall be double unless the wings are so designed that with any lift wire removed the strength of the remaining wing structure will be adequate to develop load factors of not less than 50 percent of the breaking load.

Aside from the loading conditions, the U.S Specifications contain a great number of valuable suggestions and hints for the designer. The limitation to a few carefully chosen loading conditions and especially, to the oversized wing fittings are well worth copying. The effect of the speed on the load factors is at least indirectly considered by the dependence on the power loading. The extremely low loading conditions for control surfaces which, in addition, are independent of the speed, have lately caused tail-surface failures. Its origin lies in previously preferred contours with poor aspect ratio, which aerodynamically were inferior.

The above Air Commerce Regulations were revised, effective January 1, 1931. They differ in some points from those of 1928 and complement them (table XXIII).

The load factors for airplanes whose gross weights lie between 1,132 kg (2,500 lb.) and 5,662 kg (12,500 lb.), will be found by linear interpolation. Asymmetrical stresses in the wing cellule must be allowed for in all loading conditions except nose dive, so that the load on one wing half is reduced to 70 percent.

Ailerons and vertical tail surfaces shall be designed for three fourths of the horizontal tail-surface load. In horizontal tail surfaces only half the load is to be assumed as acting upward. When auxiliary wings (slotted wings) are used, a unit loading of 366 kg/m^2 perpendicular to the chord of the slot is required.

The load on the wing ribs in case A shall be distributed conformably to table XXIV.

Table XXIII. U. S. Load Factors, January 1931

Gross weight of airplane (t)	≤ 1.13	2.27	6.80	≥ 11.34
Landplanes, multi-engine	6.5	5.0	4.5	4
single-engine, $\frac{G}{N} \geq 8.95$ kg/hp	6.5	5.0	4.5	4
single-engine, $\frac{G}{N} \leq 2.25$ kg/hp	8.5	6.4	5.45	5
Seaplanes and amphibians, multi-engine	5.85	4.75	4.5	4
single-engine $\frac{G}{N} \geq 8.95$ kg/hp	5.85	4.75	4.5	4
single-engine, $\frac{G}{N} \leq 2.25$ kg/hp	7.65	6.08	5.45	5
Load on horizontal tail surfaces in kg/m² for				
multi-engine types, $\frac{N}{F} \geq 31.1$ hp./m²	269	171	122	122
single-engine types, $\frac{N}{F} \geq 21.8$ hp./m²	269	171	122	122
all airplanes, $\frac{N}{F} = 0$	73	73	73	73

Table XXIV. Rib Loading, Case A

Wing chord	0	0.06	0.22	0.6	1.0
Relative load	1.0	$\begin{cases}1.0\\0.8\end{cases}$	0.46	0.14	0.10

By medium angle of attack a triangular load distribution with a maximum not exceeding one fourth of the wing chord is required.

The load distribution over the fixed surfaces is to be assumed uniform, but on movable surfaces the intensity of loading at the hinges shall be equal to the loading on the fixed surfaces in front of them and shall decrease uniformly to an intensity at the trailing edge of one third the former value.

The control shall be designed to withstand 1.25 times the tail surface loads, but at the highest and lowest corresponding to hand and foot power given in table XXV.

Table XXV. Control Forces and Moments

Gross weight of airplane	Tail surfaces				Ailerons			
	Horizontal		Vertical					
	kg	kg	kg	kg	kg	mkg	kg	mkg
$\leqslant$ 0.23 t	136	45	136	91	57	57 d	16	46 d
$\geqslant$ 0.68 t		91					34	34 d

The skid shall be designed to sustain the height of free drop for three-point landing given in table XXVI. The rules for landing skids are the same as for landing gears with wheels. If the skids are used for purposes other than snow or ice landing, a coefficient of friction of 0.5 must be considered. In landing with braked wheels a coefficient of friction of 0.55 with two-times safety shall be assumed. For the load analysis the gross weight of the airplane, the travel in position of rest and undeflected tires are required. The Department of Commerce approves the strength of airplanes given in table XXVII.

The fuselage shall be designed to withstand, aside from the flight and landing stresses, the propeller thrust and, in the case of direct-drive engines having 2, 3, 4, 5, or more cylinders, the stresses due to torque load and its equilibrium shall be multiplied by a load factor of 6, 5, 3, and 2, respectively. In the case of geared engines, having a reduction of 3.2 or greater, the load factors shall be 9,

Table XXVI. Stress at Landing

Gross weight of airplane (t)		≤ 0.45	0.45	1.13	2.27	6.80	≥ 11.34
Vertical impact factor for landplanes	G/F = 14.6 kg/m^2	5.0	6.5	6.5	5.5	4.9	4.5
	" = 48.8 "	6.5					
Free height of drop for landplanes (m)		0.381	0.457	0.457	0.406	0.356	0.305
Vertical impact factor for seaplanes	G/F = 14.6 kg/m^2	5.0	6.5	6.5	6.5	6.5	6.5
	" = 48.8 "	6.5					
for floats and float-brace system	" = 14.6 "	6					
	" = 48.8 "	7.5	8	8	8	8	8

Table XXVII. Strength of Airplane Landing Wheels

Tire size	10 x 3	14 x 4	18 x 3	24 x 4	28 x 4	30 x 5	32 x 6	36 x 8	44 x 10	54 x 12
Inflation pressure kg/cm^2	3.52	3.52	3.52	3.52	3.52	3.52	3.87	4.22	4.57	4.92
Static load (kg)	147	181	238	386	454	726	988	1,814	2,858	4,536
Radial breaking load (kg)	-	1,361	2,268	3,175	3,856	4,990	5,919	9,072	14,969	22,680
Side " " (kg)	-	-	907	1,134	1,361	1,497	1,837	1,722	4,491	6,804

8, 6, 3, respectively. All fittings shall be designed to carry loads 20 percent in excess of the design loads for the members to which they are connected.

Loading Conditions for Gliders

The breaking load factors for gliders are:

6.0 for high angle of attack,
4.25 for low angle of attack,
2.5 for inverted flight.

The loading for the nose dive condition shall be the same as that stipulated for airplanes, except that the total of the chord components of the loads shall equal 75 percent of the gross weight of the glider. The wings shall be able to carry the glider with a load factor of 1.5 at the wing tips imposed by handling. The breaking loads to be used for the design of the control surfaces of gliders shall be 59 kg/m² (12 lb./sq.ft.), for horizontal, and 44 kg/m² (9 lb./sq.ft.) for vertical tail surfaces and ailerons. The strength of the control system shall not be less than required to carry 1.25 percent of the maximum air loads on the various control surfaces.

Gliders having a landing speed greater than 9 m/s (20 m.p.h) shall be provided with skids or wheels, and have a load factor 5 at the minimum. The landing conditions are the same as for airplanes. For seaplane gliders, the load factor required shall be 5.

Safety belts shall be designed for 386 kg (850 lb.) breaking load.

These new strength requirements stress the effect of engine power on the magnitude of the stresses more than heretofore. It would have been better, even though a little more inconvenient in the design, if there had been a direct reference to the maximum horizontal speed, since the flight performances used as basis can be quickly out-distanced by aerodynamic and automotive improvements.

Amendments to these regulations were not long in coming as evidenced by the issuance of the August 1931 bulletin (reference 59, table XXVIII).

The allowances for multi-engine airplanes are voided in the future. All external wire bracing shall have a

safety factor 2 as against the other members of the structure; the ribs, a safety factor 1.2.

Table XXVIII. U. S. Load Factors, August 1931

Gross weight of airplanes (t)	≤ 1.134	2.268	6.804	≥ 11.34
Landplanes,				
$\frac{G}{N} \geq 8.95$ kg/hp	6.5	5.	4.5	4
5.45 "	7.55	5.75	5.0	4.25
≤ 2.25 "	10.4	8.4	6.95	5.5
Seaplanes and amphibians,				
$\frac{G}{N} \geq 8.95$ kg/hp	5.85	4.75	4.5	4
5.45 "	6.80	5.46	5.0	4.25
≤ 2.25 "	9.36	7.98	6.95	5.5

The stress of the horizontal tail surfaces in the nose-dive conditions is decided by the loading given in table XXIII, instead of the front spar load.

The severe regulations about the magnitude of the lateral forces and the decelerating forces of the landing gear were somewhat loosened. When low-pressure tires are used the bending loads for axles and landing-gear struts may be reduced by 35 percent.

A promising step in the above cited sense toward the rationalization of load factors for wings and control surfaces is found in J. S. Newell's paper "Preliminary Study of Load Factor Determination," (reference 60).

The stress in a sharp pull-out (case A) is computed from the maximum horizontal speed at ground level, according to the approximation formula

$$v_h = \sqrt[3]{\frac{N}{F}} \qquad (43)$$

with $C = 26.4$ for monoplanes and $C = 25.6$ for biplanes. To account for the effect of various modifications in design the following factors can be added:

cantilever or wire-braced wings	1.04
oval fuselage	1.02
monocoque fuselage	1.04
retractable chassis	1.07
multi-engine airplanes	0.95
seaplanes, flying-boats	0.96

The breaking-load factor becomes

$$n_A = k \left(\frac{v_h}{v_l}\right)^2 \geq 4.2\ k \tag{44}$$

Figure 41 shows k for several U. S. airplanes. The recommended mean value is

$$k = 0.85 + \frac{1910}{1820 + G} \tag{45}$$

The load factor at low angle of attack (case B) is based upon the terminal velocity in the nose-dive condition for which the range $v_e = 1.6 \div 2.6\ v_h$ is given.

By approximation

$$v_e = 1.61\ v_h \sqrt{k} \tag{46}$$

which is considered ample according to an investigation of different spar arrangements, when the breaking-load factor is

$$n_B = 0.25 \left(\frac{v_e}{v_l}\right)^2 \tag{47}$$

so that

$$n_B = 0.65\ n_A$$

It looks as if the far-reaching simplifications were introduced to explain an otherwise well-known relationship.

In the nose-dive conditions (case C) the breaking strength shall be

$$p_{HF} = 1.20\ p_H = 1.2\ c{m_o}' \frac{t}{l_{H'}} \frac{F}{F_{HF}} q_e \tag{48}$$

The distribution over the stabilizer chord is such as to act in stages of

$1.71\ p_{HF}$	from	0	to	$0.15\ t_{HF}$
$1.14\ p_{HF}$	"	0.15	"	$0.60\ t_{HF}$
$0.57\ p_{HF}$	"	0.60	"	$1.00\ t_{HF}$

The elevator remains relieved. The factor 1.2 is assumedly to be the factor of safety.

For pull-out (case A) the unit load on the horizontal tail surfaces is

$$p_H = 1.17\ k\ \frac{F_{HR}}{F_H}\ q_h \tag{49}$$

The distribution over the elevator chord is triangular with value $2\ p_H$ over the elevator axis and linearly decreasing to zero toward leading and trailing edge.

Translation by J. Vanier,
National Advisory Committee
for Aeronautics.

For Part III, see N.A.C.A. Technical Memorandum No. 718, which follows.

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Clarke, T. W. K.: The Maximum Loading Attainable on the Wings of an Airplane Owing to the Flattening-Out after a Prolonged Dive. R. & M. No. 128, British A.C.A., 1914, p. 353.

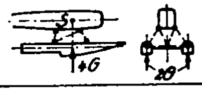
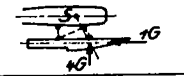
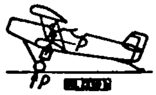
Case	Safe load factor: Vertical	Safe load factor: Horizontal	Safe load factor: Transverse	Type of loading
1	4			
2	4	1		
3	4		1	
4	2			

Figure 35.-Loading conditions for floats (1926)

1a) 3 point landing

P=eG$\leqq$3.0G

1b) Wheel landing
Skid at instant of touching the ground

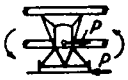


P=eG$\leqq$3.0G

2) 1 wheel landing
Skid at instant of touching the ground

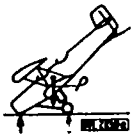
P=0.33eG$\leqq$1G

3) Side load



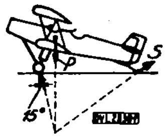
P=0.15eG$\leqq$0.45G

4a) Nosing over



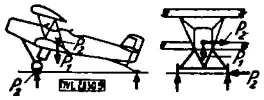
P=2.50G

4b) Dragging



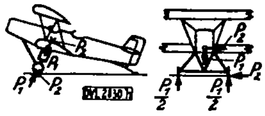
P=1.50G

Case 1a+3
Combined with 3/4 of individual loads



P_1=0.750eG$\leqq$2.25G
P_2=0.113eG$\leqq$0.225G

Case 1b+3
Combined with 3/4 of individual loads

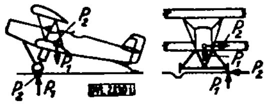


P_1=0.750eG$\leqq$2.25G
P_2=0.113eG$\leqq$0.225G

6a) Skid impact vertical corresponding to 1a
6b) Skid impact from the side

$P=0.15e_{sp}G_{sp}$
$(e_{sp}>3.0)$

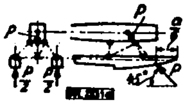
Case 2+3
Combined with full individual loads



P_1=0.33eG$\leqq$1.0G
P_2=0.15eG$\leqq$0.3G

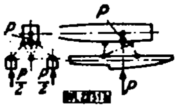
Figure 36.-Loading conditions for landing gears (1927)

Ia) Bow landing



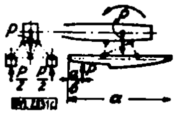
P=eG

Ib) Step landing



P=eG

Ic) Stern landing

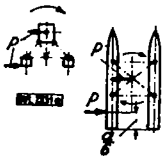


P=0.5eG

IIa to c) Side landing

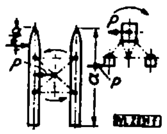
Position of resultants
as for Ia, Ib, and Ic.
In case:
IIa:P=0.5eG
IIb:P=0.5eG
IIc:P=0.25eG

IIIa) Nose landing



P=0.25eG

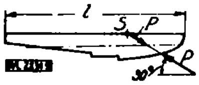
IIIb) Stern landing



P=0.25eG

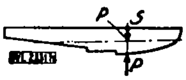
Only in combination with I, each 3/4 of individual loads.
Only in combination with II, full individual loads.

Ia) Bow landing



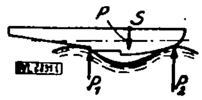
P=eG

Ib) Step landing



P=eG

Ic) 2 wave landing



P=eG

II) One sided landing

P=0.5eG
b: width of wing tip floats and stub

III) Side landing

P=0.33eG
Only in combination with I, each 3/4 of individual loads.
Only in combination with II, full individual loads.

Figure 37.-Loading conditions for float seaplanes and flying boats (1927).

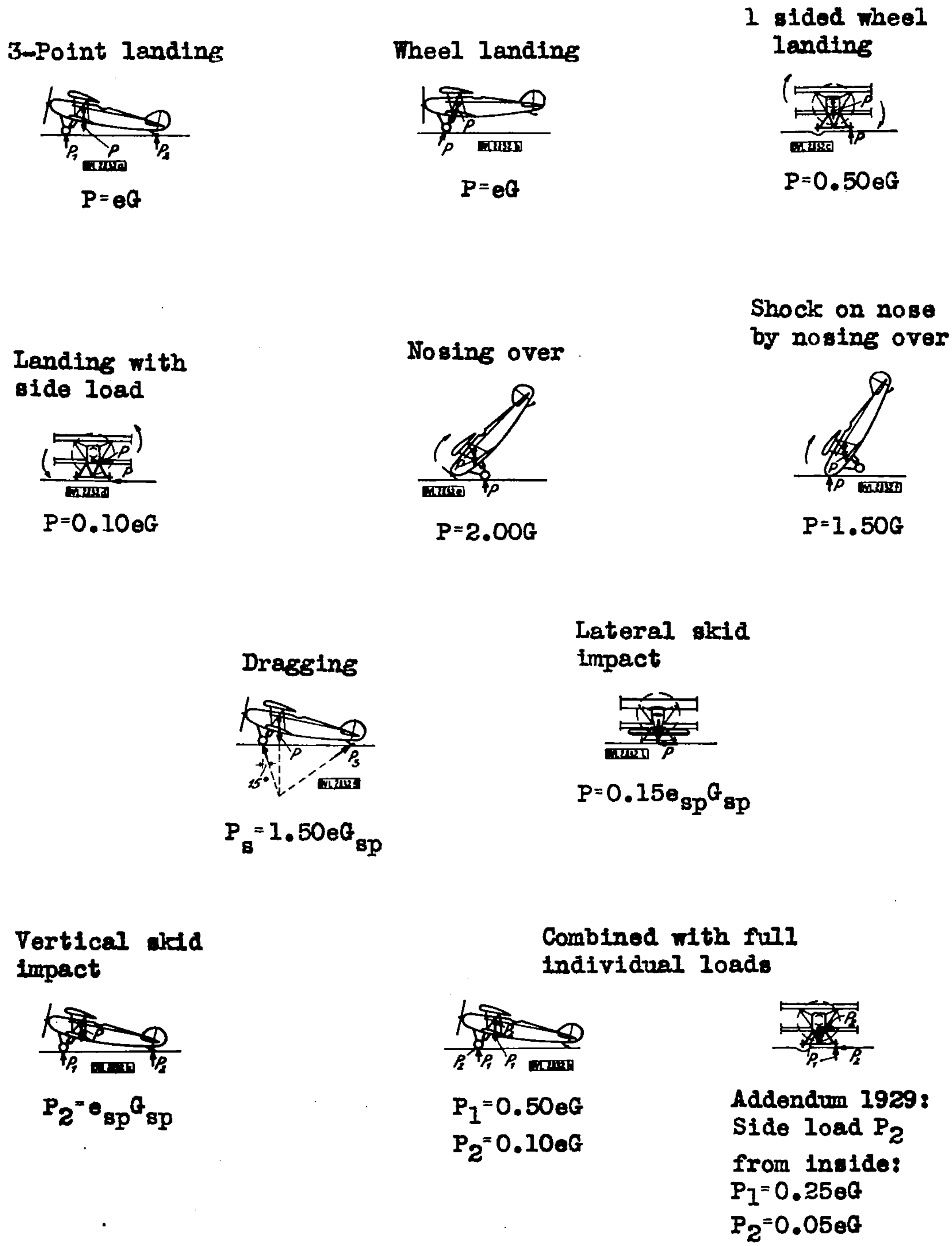
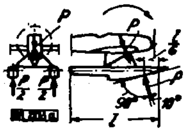


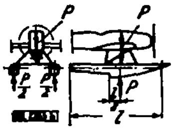
Figure 38.-Landing conditions for landing gears (1928/1929)

Ia) Bow



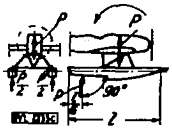
P=0.7eG
Length of floatation line

Ib) Step



P=eG

Ic) Landing on stern

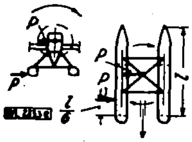


P=0.4eG

IIa to c) One side landing

Position of resultant as la, lb, and lc.
In case:
IIa: P=0.35eG
IIb: P=0.5eG
IIc: P=0.2eG

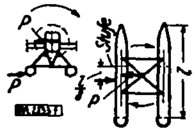
IIIa) Side landing-bow



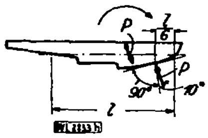
P=0.12eG
(only combined with IIa)
Amended 1929:
P=0.1eG

Direction of flight

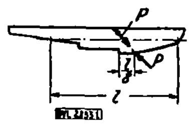
IIIb) Side landing-step



P=0.25eG
(only combined with IIb)
Amended 1929:
P=0.15eG

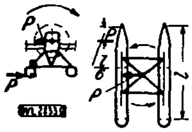


P=0.7eG

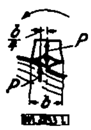


P=eG

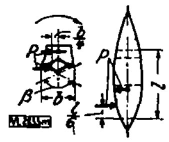
IIIc) Side landing-stern



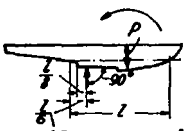
P=0.08eG
(only combined with IIc)
Amended 1929:
P=0.06eG



Position of resultant as Ia, Ib, and Ic.
b: width of wing tip floats and stub.
In case:
IIa: P=0.35eG
IIb: P=0.5eG
IIc: P=0.2eG

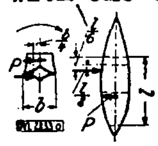


P=0.18eG
(only combined with IIa)

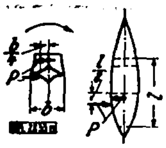


Or with one step
P=0.4eG

Or with one step

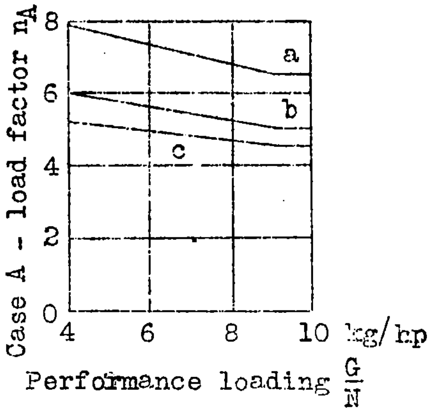


P=0.10eG
(only in combination with IIc)



P=0.35eG
(only combined with IIb)
Amended 1929:
P=0.25eG

Figure 39.-Loading conditions for float seaplanes and flying boats (1928/1929).



a) gross weight $\leqslant$ 1132 kg (2500 lb.)
b) " " = 2265 " (5000 lb.)
c) " " $\geqslant$ 5662 " (12500 lb.)

Figure 40.-Ultimate load factor of U.S. airplane in case A.

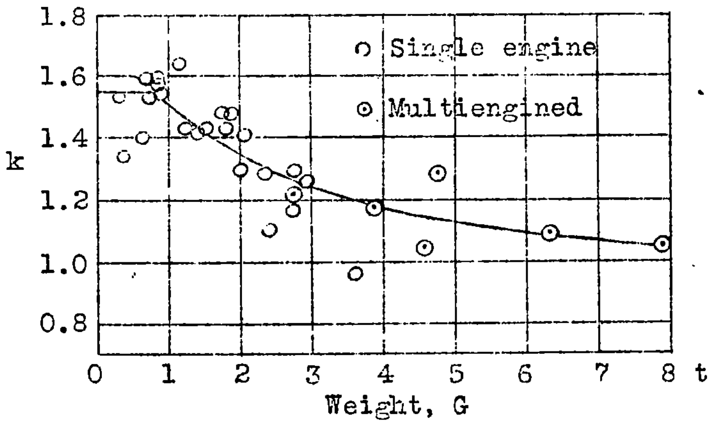


Figure 41.-Ratio k = q_A failure. q_h for various U.S. airplanes (source U.S. Air Commerce Bulletin, Vol.3, (1931), No. 9, Fig. 5).